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IMP J

SUMMARY TECHNICAL

REPORT

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EMR-Aerospace Sciences
EMR Division
Weston Instruments, Inc.
5012 College Avenue
College Park, Maryland 20740

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SECTION I

INTRODUCTION

The IMP J spacecraft was launched from Complex 17B of the Eastern Test Range at 2226 EDT on 25 October 1973 (0226, 26 October, GMT). Upon successful launch, IMP J was designated Explorer 50.

The IMP J Summary Technical Report is published by EMR-Aerospace Sciences to provide technical and administrative guidance for design, fabrication, integration, testing, and prelaunch activity on future spacecraft programs. General problems encountered on the IMP J program are described, and a recommendation is offered for these problems on future programs.

This report addresses technical problems encountered during the IMP J program which could recur on future programs; it is not the intent of the report to criticize program administrative policy or program "philosophy".

Each of the pertinent problems which occurred during the IMP J program is generally described in this report. This description is followed by a summary conclusion as to the nature of the problem including a statement of the conditions which caused the problem to occur. In each case, a recommendation is given for avoiding similar problems on future spacecraft programs.

SECTION II

DESIGN PROBLEMS AND RECOMMENDATIONS

The following design problems were encountered and are noted here because they are potential problem areas on future spacecraft.

2-1. ELECTRICAL CONNECTOR MISMATING

The IMP J spacecraft had approximately 100 electrical connectors. A few locations existed where a connector could be incorrectly mated. One location concerns the Turn-On Plug which is physically, but not electrically, interchangeable with the EED Arming Plug. Another location concerns the two pyrogen connectors which appear the same as the connector for the kick motor pressure transducer.

2-1-1. Conclusion

Physical mating of connectors at incorrect locations can cause damage to the spacecraft or personnel.

2-1-2. Recommendation

Optimally, non-fixed connectors should be unique physically. This may not always be possible. However, this type problem should be recognized early. Then the appropriate color coding could be used to minimize incorrect connector mating. Frequently, equipment designs are complete before this type problem can be recognized. Thus, the color coding solution would be the most readily implementable. The color coding should be unique and applied to both the non-fixed and fixed connectors or the area adjacent to the fixed connector.

2-2. SPACECRAFT BATTERY PROTECTION

The spacecraft battery was not required to deliver more than $\approx$ 8 amps steady-state or $\approx$ 20 amps transient. However, the 14-cell AgCd, 10 AH battery could easily deliver 100 amps for a short period of time.

In order to minimize the IR drop from the battery terminals, four #20 wires were tied in parallell in the harness. In this configuration, a short circuit in the spacecraft harness from $+V_b$ to $-V_b$, signal common, 28V return, or chassis could cause the spacecraft harness insulation to burn-off with resultant damage to other parts of the harness.

Depending on several parameters, some other point in the circuit may open circuit before significant harness damage occurs. This, however, is dependent on and varies with the situation.

2-2-1. Conclusion

The spacecraft battery is unprotected. This is particularly true when the spacecraft is not powered-up since the level detectors cannot then function to protect the battery. Even with the spacecraft powered-up, the level detectors only open the battery switch. This would not necessarily open a short circuit since the switch is not physically near the battery, but is inside the System Programmer.

2-2-2. Recommendation

To protect the spacecraft battery or to protect the spacecraft wiring harness from the battery, the ability of the battery to deliver current should be limited. This limiting may take several forms and could also be varied for "ground operations" versus in-orbit operations.

It is recommended that for ground operations, a fuse or fusible link (short piece of wire that will fuse at approximately 20 amperes in 50 msec) be used on future spacecraft. This link would be at the battery terminals. For launch, the link could remain as is or be jumpered with a wire or switch of additional current capability. If jumpered with a switch, this could be a commandable latching relay or a ground-only stimulated relay.

This seems to be a proper area for additional consideration and investigation. Coupled into this consideration could be the possibility of firing EED's with capacitors properly sized so that the battery is not essential to EED firing. Preliminary calculations show that for $\approx$ 12 amperes and 5 msec, a capacitor of the order of 3000 μf would be required.

2-3. SPACECRAFT CURRENT

The spacecraft current (I_{SC}) is read out in telemetry every 40.96 seconds with a resolution of ± 35 ma.

2-3-1. Conclusion

There are advantages to having I_{SC} read out more frequently and with better resolution. For example, some spacecraft functions, when commanded, cannot be monitored by I_{SC} since the ΔI_{SC} is within the established resolution.

2-3-2. Recommendation

When advanced data systems on the spacecraft are operational, it would be desirable to read out I_{SC} more frequently in a variable format and with improved resolution (± 3 ma).

2-4. RF TRANSPARENT DETECTOR COVERS

The IMP H Summary Report, Section 2-5, page 2-3, recommended that experiment detector covers be RF transparent in order to expose the experiments to RF rather than protect them from RF.

2-4-1. Conclusion

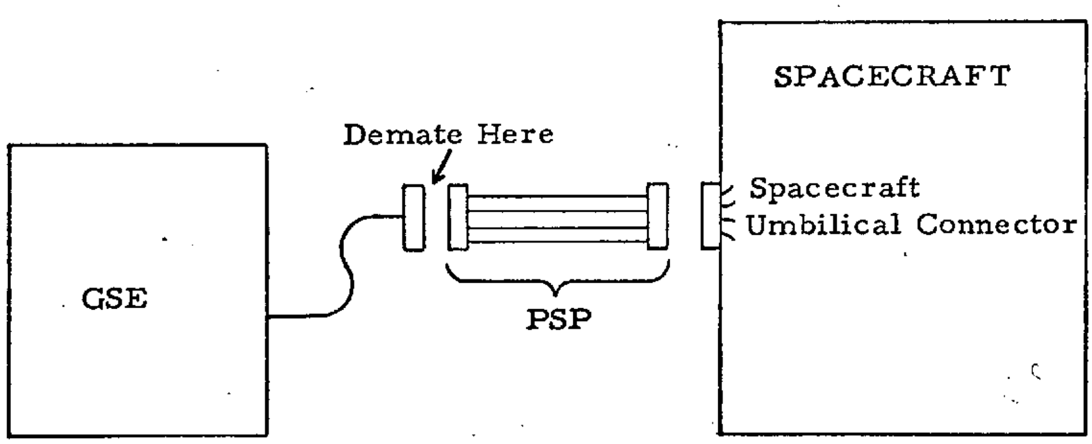
This recommendation from the IMP H Summary Report was implemented for IMP J. Then RF interference problems were noted and solved early in the program.

2-4-2. Recommendation

Future spacecraft should be carefully reviewed for the value which may be obtained by using RF transparent covers instead of covers which would protect experiments from RF interference.

2-5. SPACECRAFT CONNECTOR PROTECTION

Some electrical connections to the spacecraft were mated and demated several times a day. This handling over a period of time can degrade the spacecraft mounted connector. Specifically, the umbilical and GSE test connectors were exercised regularly. To avoid damage to these connectors, a short extension, designated Pin Socket Protector (PSP), was attached and left connected to the spacecraft connector. The make and break connection was then secured at the non-spacecraft end of this short extension. Thus, any connector damage would be confined to the PSP.



2-5-1. Conclusion

The PSP protection of spacecraft connectors saved some rework time since it was anticipated that the excessive mating and demating would have degraded the spacecraft connectors.

2-5-2. Recommendation

For future spacecraft, an evaluation should be made of those areas where the PSP could effectively be employed. However, occasional testing without the PSP's should be performed to ensure that the connection without PSP's works properly.

2-6. DEUTSCH CONNECTORS

Schedule time was lost since Deutsch harness connectors had to be individually shimmed to maximize pin engagement.

2-6-1. Conclusion

Tolerances on Deutsch connector dimensions were sufficiently loose that intermittent electrical contact resulted. Since these connectors rely on the position of the instrument and harness panel to assure full pin engagement, deviations from the established tolerances can produce intermittent connector pin contact. The loose tolerances coupled with flexing of honeycomb panels resulted in commands not being received by the spacecraft during ground operations.

2-6-2. Recommendation

The following recommendations are offered to eliminate the problem:

a. Minimize the use of Deutsch connectors in a "rack and panel" installation on the spacecraft.

b. Provide better definition of Deutsch connector dimensions and tolerances in order to maximize pin engagement. Current pin engagement for Deutsch connectors is approximately 0.055-inch as compared to 0.125-inch for Cannon connectors.

c. Avoid the use of Deutsch connectors where low pin density is a factor.

2-7. CAPTIVE SCREWS

Schedule time was lost on IMP J searching the spacecraft for screws dropped during installation of flight plugs. Although no problem was encountered at ETR, a screw dropped inside the fairing or on the gantry would have caused loss of time.

2-7-1. Conclusion

Captive screws were not used on all spacecraft components removed from the spacecraft on the gantry.

2-7-2. Recommendation

All experiments and instruments must have their removable components equipped with captive screws when the cards are delivered for integration. This includes sensor protective covers and enable and disable plugs.

2-8. SPACECRAFT RF CONFIGURATION

When testing the spacecraft for RF interference either in an anechoic chamber or in a laboratory, it was difficult to configure the spacecraft sufficiently close to the orbital situation to make the test valid. Although some areas must be compromised due to lack of experiments or flight solar panels, etc., other areas can and should be configured more nearly orbital. For example, removal of solar panel covers, deployment of booms, and installation of experiment cover panels.

2-8-1. Conclusion

RF interference testing is frequently misleading due to the RF environment or the spacecraft RF configuration.

2-8-2. Recommendation

Spacecraft configuration and environment should be reviewed more closely before RF interference tests since these parameters affect the test results and confuse the test conclusions.

A scheme for resolving the RF environmental aspect of the problem was partially implemented at Hangar S of ETR for the IMP J prelaunch operations. This scheme used an RF detector probe, placed near experiments, to determine their RF levels when the spacecraft was operational and using flight antennas. In this manner, the spacecraft could be mapped for direct and reflected RF at any point. It is specifically recommended that this idea be further developed as a tool for determining whether RF interference at an experiment is real or due to a poor RF environment.

As a starting point, a mapping probe would be designed, developed, and used with a spacecraft in the optimum environment of an anechoic chamber to map the spacecraft areas. The spacecraft would then be remapped during subsequent RF testing in a lab or elsewhere; and this data would be compared to the anechoic chamber data to determine if the RF interference at an experiment is due to the uncontrolled RF environment.

2-9. OPTICAL ASPECT SYSTEM TESTING

The test set-up and instrumentation for dynamically testing the OA system in the spacecraft is insufficient for the task, and has resulted in additional questions upon test completion. On smaller spacecraft (IMP F and G), the spacecraft was tested outside in sunlight. With larger spacecraft, this is not as easily performed.

Areas of responsibility for this test are defined less clearly than for any other similar spacecraft tests. This relates mostly to test instrumentation and techniques. A prime problem is that the lamp used to stimulate the OA sensor may work marginally or not at all. The lamp parameters of intensity, spectral purity, collimation, and alignment to the sensor have not been sufficiently under control for the last several tests.

2-9-1. Conclusion

The test set-up for the dynamic OA system test lacks a proper and controlled stimulating light source.

2-9-2. Recommendation

For subsequent spacecraft OA system tests, the stimulating lamp should be improved as related to the spectral output, intensity, collimation, and alignment of the lamp to the spacecraft sensor. Either T&E or the OA Instrumenter should be responsible for this test equipment, but not both.

2-10. OPTICAL ASPECT SYSTEM GEOMETRY

On both IMP H and IMP J there was confusion up until launch regarding the geometric relationship between the Optical Aspect sun slit and earth telescope. In order to determine the earth telescope line-of-site relative to the spin axis and sun slit, certain rough measurements were made on both IMP H and J spacecraft at ETR.

2-10-1. Conclusion

The geometry associated with the Optical Aspect system was not understood completely by anyone except the OA Instrumenter.

2-10-2. Recommendation

At the start of spacecraft design, a review should be held during which all requirements for attitude sensing equipment is discussed in detail. If the line-of-site of sun and earth sensors is quite critical, then a special rigid structure should be provided for that system. In addition, the instrumenter should write a test procedure for determining these LOS, and he should provide the associated GSE.

2-11. PROJECT DRAWINGS

On both the IMP H and IMP J spacecraft, the spacecraft umbilical connector was keyed 180° differently than the mating connector on the fairing provided by MDAC. Consequently, the fairing connector and pigtail had to be rotated to mate with the spacecraft. This caused unnecessary stress on the wiring.

2-11-1. Conclusion

The Delta Interface Document generated by the spacecraft personnel and the Compatibility Drawing generated by the Launch Vehicle Contractor were not thorough concerning all details.

2-11-2. Recommendation

The Project Office should review these documents for details, such as connector keying and lanyard material, length, and positioning.

2-12. SUBSYSTEM FIT CHECKS

On several occasions during the IMP H and J program, an existing problem was discovered just prior to starting a certain operation or test. Some of these problems were physical interference, cable routing, lack of material, wrong size material, etc. As a result, either schedule time was lost or the test configuration was compromised.

2-12-1. Conclusion

In the interest of meeting production and test schedules, full fit checks of certain subsystems were compromised. Included in this category were: 1) fully configured experiment booms, and 2) fully configured kick motor, including thermal blankets.

2-12-2. Recommendation

A formal fit check must be required and witnessed by the Project Office with photographic coverage. These photos will help to answer subsequent questions, and the fit checks will assure that tests may start on schedule.

SECTION III

INTEGRATION PROBLEMS AND RECOMMENDATIONS

The following problems were encountered during integration and are noted here because they are potential problem areas on future spacecraft.

3-1. DELIVERY OF COMPONENTS

To integrate a spacecraft, a prescribed order is followed: a) structure, b) harness, c) power system, d) telemetry system. After this point, any of several paths may be chosen. During integration of IMP J, sometimes components were not available and compromise paths of integration were taken. For example, several components of the power system were integrated, but they were not intended for flight or as spares.

3-1-1. Conclusion

Deviations from a planned order of integration create difficulties regarding uncompleted tests and procedures.

3-1-2. Recommendation

Scheduling of instruments should be more closely related to the order of integration. Compromise routes should be avoided since they introduce risk to the spacecraft.

3-2. INSTRUMENTATION TECHNIQUES

This is an area where several problems were avoided and is mentioned here so that these techniques may continue to be used on future spacecraft.

Instrumenting to a component or the spacecraft harness was performed to virtually eliminate an accidental short circuit. For example, oscilloscopes were always grounded through the power cord, and the oscilloscope chassis was always insulated from the spacecraft. A one-to-one scope probe was used, but it had a built-in 1 k-ohm resistor in series. Thus, an accidental short of the cable at the oscilloscope end would be through 1 k-ohm and, in most cases, it would cause no damage.

Oscilloscope measurements were made without referencing the oscilloscope to spacecraft signal common. The oscilloscope was only referenced through its power cord, and ultimately to spacecraft signal common. This technique can avoid many possible short circuit problems.

3-2-1. Conclusion

Instrumentation techniques employed on IMP J avoided damage to the spacecraft through accidental shorts, etc. This indicates that good techniques will prevent problems.

3-2-2. Recommendation

Instrumentation techniques on the spacecraft should be continuously reviewed and updated from the point of view of protecting spacecraft systems.

3-3. DEFECTS IN DELIVERED COMPONENTS

Components delivered for inspection and integration frequently do not pass the incoming inspection. When this occurs, a Discrepancy Report is written and the component returned to the Instrumenter or Experimenter for repair or modification.

3-3-1. Conclusion

Delivered items which do not pass inspection cause schedule delays and generate additional paperwork.

3-3-2. Recommendation

A careful visual inspection of components at GSFC prior to delivery to EMR for Receiving Inspection could reduce rejections by as much as 50 percent. Thus, it is recommended that components be submitted to a thorough visual inspection by GSFC QC prior to delivery.

3-4. APPROVED MATERIALS

The integration team continually monitored the spacecraft vicinity for unapproved materials which might be harmful to solid state detectors. In addition, the materials used on the spacecraft had to be screened and approved.

3-4-1. Conclusion

Interfacing organizations were not always apprised of the importance of the Approved Materials List. This includes Experimenters and the Vehicle Contractor.

3-4-2. Recommendation

a. Integration personnel must continue to coordinate with the GSFC Materials personnel and periodically update the Approved Materials List.

b. An updated copy of the Approved Materials List must be available to the Test Conductor.

c. The Approved Materials List should be distributed to all interfacing organizations on a periodic basis.

SECTION IV

ENVIRONMENTAL TEST PROBLEMS AND RECOMMENDATIONS

The following problems were encountered during the Test and Evaluation (T&E) phase at GSFC and are noted here because they are potential problem areas on future spacecraft.

4-1. THERMAL VACUUM TEST VS. HIGH VOLTAGE

During the Thermal Vacuum Tests, the possibility of high voltage discharge from an experiment is always present. Several precautions must be taken to minimize this possibility.

a. Have a detailed Design Review of the experiment high voltage design with particular attention to packaging techniques,

b. Provide a clean vacuum chamber (relatively free from outgassing),

c. Provide a clean spacecraft (relatively free from outgassing),

d. Establish a sufficient time-pressure factor before high voltage turn-on (i.e., 10 hours at 1×10^{-6} torr or less before turn-on).

4-1-1. Conclusion

To avoid a high voltage discharge problem in thermal vacuum, the previous precautions must be followed. The vacuum chamber cleanliness, spacecraft cleanliness, and time-pressure factor are presently recognized and agreed upon principles which have been applied to IMP J with success.

The remaining item is a high voltage design review with particular attention to packaging techniques, and it requires greater discussion. The high voltage design review of experiments should be held separate from the general review and should be conducted by a GSFC high voltage packaging expert. This review should be held early, i.e., before significant packaging design has been performed. It would be most desirable to have examples of high voltage power supplies available for display at this design review. Having a list of materials and examples would also be of value. Displaying an X-ray of a hollow core resistor and an X-ray of a resistor without a hollow core, along with cut-open examples of each resistor type, could be valuable.

4-1-2. Recommendation

Although IMP J experienced no high voltage problems in thermal vacuum, careful consideration of the previous precautions is recommended for future spacecraft. A definite trend of less and less high voltage problems has been established when IMP F, G, I, H, and J are considered, with IMP J having had no high voltage problems during the thermal vacuum testing.

The high voltage design review should be held separate from the general design review and should include those persons who are associated with high voltage design and packaging problems. Since it may be advisable to have this review organized and conducted by a GSFC high voltage expert, this is being recommended. The high voltage review should display physical examples of good and bad component selection and packaging techniques.

Even with the considerable effort being applied toward solving the high voltage vs. thermal vacuum test problems, many packaging designs seem oblivious to the prior history of problems. There is much information in existance concerning these high voltage problems. However, one of the reasons that all this data seems to be ignored is that it is not compiled into a more comprehensive form. It is recommended that such data be compiled into a guide book and be made available to designers.

Precautions b, c, and d listed at the beginning of this section, and requiring a clean vacuum chamber, a clean spacecraft, and a sufficient time-pressure factor, have been implemented for IMP J. However, continued attention should be paid to these areas. It is recommended that the vacuum chamber be "baked out" prior to installing the spacecraft for a thermal vacuum test. Two choices are available for this "bake out". One choice is to use the same temperature cycle for the empty vacuum chamber as the first cycle will be for the spacecraft (see Figure 1). An advantage of this cycle is that the pressures encountered with the chamber empty can then be correlated to the

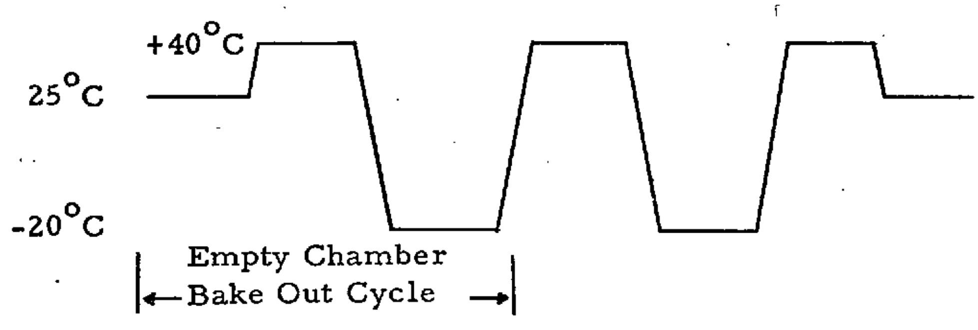


Figure 1. Vacuum Chamber Temperature Cycle

pressures with the chamber containing the spacecraft. This result could give an indication of spacecraft outgassing. The other choice is to "bake out" the empty vacuum chamber at as high a temperature as practicable and for as long as reasonable (24 hours to 48 hours).

A clean spacecraft can be obtained if certain items are properly handled. Basically, the spacecraft will be clean in some respects due to the cleanliness restraints. However, materials which will outgas under vacuum are not easily removed except by a vacuum exposure. It is assumed that this exposure removes most materials which could outgas since each component has undergone a thermal vacuum test. Some parts of the structure have not been exposed to vacuum and are the most probable contributors to a large gas load. Specifically, any honeycomb structure parts are potential outgassing areas. This would be -- main platform, harness support panels, dummy solar panels, and the live solar panels. The gas load from such honeycomb structure is unknown. It appears that the dummy solar panels outgas heavily during thermal vacuum testing. Due to this suspicion, the panels were "baked out" in a vacuum chamber prior to a spacecraft thermal vacuum test. Subsequently, it appeared that the spacecraft gas load was less. The relative cleanliness of such a structure could be determined by a special test on a sample piece and this is recommended. If it is determined that the honeycombed structure is a large contributor to outgassing, then all honeycombed pieces should be thermal vacuum exposed prior to installation on the spacecraft.

In addition to having a clean spacecraft, it is desirable to first expose the spacecraft to a hot temperature instead of a cold temperature. The hot exposure should be performed with the spacecraft experiments off.

The exposure temperature should be $\approx +40^{\circ}C$ for at least 24 hours. This will provide the spacecraft with an opportunity to outgas. If the empty chamber had been operated previously at this temperature (outgassed), then a comparison of the pressure vs. time curves for the chamber with and without the spacecraft would indicate the relative cleanliness of the spacecraft.

For IMP J, a single thermal vacuum test was performed by using the hot non-operating exposure first. No high voltage problems occurred. Therefore, it seems fair to attribute some of this success to the thermal vacuum operating modes.

A sufficient time-pressure factor in a vacuum chamber before high voltage turn-on has already been implemented for IMP J. The procedures call out exactly how many hours are required at a certain vacuum before high voltage turn-on. This factor was used on IMP J and no high voltage problems occurred.

4-2. SUBSYSTEM TESTING AT T&E

The subsystem (component) testing at T&E is conducted by a GSFC Test Conductor working with the Experimenter or Instrumenter. During this testing, there are situations where the Test Conductor should exercise his authority to stop the test, or contact the Project Office, or write a Malfunction Report (MR). Several areas have been noted where this authority has not been exercised. This in turn affects mission success.

During subsystem testing (as in spacecraft system testing), a correct positive action must occur at the proper time. If this correct action is not taken, the opportunity is lost since the action cannot be taken while the test is still in process. The problem, malfunction, or failure is then published weeks or months later in a test report. When the test data become available, the subsystem may already have been integrated into the spacecraft. At this time, a decision to reperform the subsystem test is severely limited by cost and schedule.

4-2-1. Conclusion

Correct action must be taken by the Test Conductor when problems occur during subsystem testing. Since the failure of the test is not known until the Test Conductor's report is published, a delay occurs. This is usually too late in the program.

4-2-2. Recommendation

The GSFC Test Conductors must be made aware of their authority to stop a test, to contact the Project Office, to write an MR, or to do all of these things. When a problem is noted, the Test Conductors may recommend that the Experimenter or the Project Office write an MR. If this does not occur, the Test Conductor should write the MR.

4-3. STORAGE BATTERIES (CELLS) USED WITH GROUND SUPPORT EQUIPMENT

For several test applications GSE incorporated the use of storage batteries. This includes lead-acid and carbon-zinc types.

All of these batteries have a massive capability of delivering energy. In particular, an automotive lead-acid battery can easily deliver 500 amperes for a short period. There is no need for such a capability near a spacecraft or personnel.

4-3-1. Conclusion

The hazard of storage batteries used in GSE has not been fully recognized. Several incidents with such batteries involving IMP J have been previously reported.

4-3-2. Recommendation

All GSE storage batteries or cells should be limited in their capability to deliver energy. Consider $500 \frac{\text{coul}}{\text{sec}}$ at $12 \frac{\text{joules}}{\text{coul}} \rightarrow 6000 \frac{\text{joules}}{\text{sec}}$.

The limiting of capability should be in the form of a properly sized fuse or a fusable link. This protection shall be as close to the output terminals of the battery as practicable in order that a short circuit before the fuse be minimized.

SECTION V

PRELAUNCH OPERATIONS PROBLEMS AND RECOMMENDATIONS

The following operation problems which occurred should be avoided in future spacecraft or launch operations.

a. The Long Functional Test at ETR was originally scheduled as five 10-hour days. Although the test was completed in five days, the time periods were considerably longer than 10 hours. This was due to the performance of many unplanned tests, which required additional time. On subsequent operations, the plan should be reviewed more closely and the time required should be estimated more realistically.

b. It would be desirable to take an Analog Tape Recorder to the launch site so that spacecraft data may be sent to GSFC for test purposes without the necessity of turning on the spacecraft. This was implemented for IMP J.

c. The continuity of data lines from ETR to GSFC were a problem at times. The lines were frequently disconnected when not in use, then they were not available when needed again. This could be avoided by personnel attaching tape to the connections and marking the use of the cable.